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RESEARCH MEMORANDUM

WIND-TUNNEL INVESTIGATION OF A SHIELDED TOTAL-PRESSURE TUBE AT A MACH NUMBER OF 1.61

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

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SUMMARY

The variation of the total-pressure error with angle of attack of a shielded total-pressure tube has been determined for angles of attack from 0° to 60° at a Mach number of 1.61 in the Langley 4- by 4-foot supersonic pressure tunnel.

The tests showed that the total-pressure error was zero up to an angle of attack of 34°, increased to $0.035q_c'$ (where q_c' is the indicated impact pressure) at 48°, then began to decrease, and reached a value of zero at 55° and $-0.07q_c'$ at 59.5°. The critical angle (the angle of attack at which the total-pressure error becomes $\pm 0.01q_c'$) was 40°.

A comparison of the results of the present tests with previous tests of the same tube at subsonic and transonic speeds showed that the critical angle remained constant at about 63° over a Mach number range of 0.26 to 0.50, decreased to 56° at a Mach number of 1.10, and then decreased rapidly to 40° at a Mach number of 1.61.

INTRODUCTION

The development of high-speed airplanes capable of maneuvering to high angles of attack at high speeds has brought about the need for fixed or rigid total-pressure tubes which will measure total pressure correctly over wide ranges of angle of attack and Mach number. In order to meet this need, the National Advisory Committee for Aeronautics is conducting a series of wind-tunnel investigations to determine the variation of total-pressure error with inclination of the air stream for a number of total-pressure tubes at subsonic, transonic, and supersonic speeds.

A preliminary investigation of 39 total-pressure tubes at subsonic speeds over an angle-of-attack range of ±45° was reported in reference 1. Subsequent tests of 20 of these tubes over an angle-of-attack range of -15° to 45° at several supersonic speeds were presented in reference 2. These investigations showed that a Kiel type shielded total-pressure tube provided the greatest range of insensitivity to inclination. In order to determine whether the range of insensitivity of this type of tube could be extended, six variations of the Kiel design tube were tested at subsonic speeds (ref. 3). These tests showed that the range of insensitivity could be increased by about 50 percent by the use of a curved-shield entry. Transonic tests of this tube were subsequently made (ref. 4).

As an extension of the investigation the shielded tube with the curved shield entry was tested at a Mach number of 1.61. This paper presents the results of this test.

SYMBOLS

H'	stagnation pressure measured by total-pressure tube at $\alpha = 0^{\circ}$
H''	stagnation pressure measured by total-pressure tube at $\alpha > 0^{\circ}$
ΔH	total-pressure error, $H'' - H'$
p_s	free-stream static pressure
q_c'	indicated impact pressure, $H' - p_s$
M	Mach number
α	angle of attack
α_{cr}	critical angle of attack, angle where $\frac{\Delta H}{q_c'} = \pm 0.01$

APPARATUS AND TESTS

A diagram of the shielded total-pressure tube is shown in figure 1. This tube is a modification of the Kiel design reported in reference 5. The present design differs from that of the Kiel tube in that the shape of the shield, or venturi entry, is curved rather than conical and the interior of the shield is vented along the walls of the tube instead of directly to the rear. This latter innovation was adapted to permit end-mounting of the tube on a horizontal boom; this mounting thus avoided the vibration difficulties which had been encountered with the spindle-type mounting of the Kiel design.

The tests were conducted in the Langley 4- by 4-foot supersonic pressure tunnel at a Mach number of 1.61 and at stagnation pressures of 6 and 15 psi. These stagnation pressures corresponded to Reynolds numbers of 1.3×10^5 and 3.3×10^5 which are based on the free-stream velocity and the diameter of the tube. The tube was mounted on a wall-support system (fig. 2) constructed so that the total-pressure entry remained in approximately the same position in the air stream for all angles of attack. The angle of attack was varied by means of a hand-operated worm-gear drive. The absolute angle was determined by means of an inclinometer, and the change in angle was indicated by a vernier attached to the cranking mechanism.

Air loads produced no measurable deflection of the support mechanism. Estimated accuracies are as follows:

	Stagnation pressure of –	
	15 psi	6 psi
$\Delta H/q_c'$	± 0.0027	± 0.0068
α	$\pm .2^\circ$	$\pm .2^\circ$

RESULTS AND DISCUSSION

The results of the tests at a Mach number of 1.61 are presented in figure 3 as a plot of the ratio of total-pressure error to indicated impact pressure against angle of attack.

As shown by figure 3 the total-pressure error is essentially zero up to an angle of 34°, increases to about $0.035q_c'$ at 48°, then begins to decrease and reaches a value of zero at 55° and $-0.07q_c'$ at 59.5°.

Variations in stagnation pressure from 6 to 15 psi (or Reynolds numbers from 1.3×10^5 to 3.3×10^5) are shown to have a negligible effect on the calibration.

This calibration differs from those obtained at subsonic and transonic speeds in one important respect. At the lower speeds the total-pressure error remained zero up to an angle of attack just below the critical angle. Beyond this point the errors became increasingly negative. The present calibration, on the other hand, shows the error to become positive before the angle of attack at which the error becomes negative. It is believed that the positive errors at angles of attack between 34° and 55° result from the fact that at α above 34° the tube no longer measures the total pressure behind a normal shock but measures instead the pressure behind an oblique shock, with resulting higher total pressure.

As can be seen from figure 3, the critical angle, the angle at which the error is $\pm 0.01q_c'$, is 40°. The critical angle for the present tests together with the critical angles at $M = 0.26$ to $M = 1.10$ (refs. 3 and 4) is plotted in figure 4 as a function of Mach number. This figure shows that the critical angle is about 63° from $M = 0.26$ to $M = 0.50$, decreases to 56° at $M = 1.10$, and then decreases to 40° at $M = 1.61$.

CONCLUDING REMARKS

Supersonic wind-tunnel tests of a shielded total-pressure tube having a curved-shield entry have been conducted through an angle-of-attack range of 0° to 60° at a Mach number of 1.61.

The tests showed that the total-pressure error was zero up to an angle of attack of 34°, increased to $0.035q_c'$ (where q_c' is the indicated impact pressure) at 48°, then began to decrease, and reached a value of zero at 55° and $-0.07q_c'$ at 59.5°. The critical angle (the angle of attack at which the total-pressure error becomes $\pm 0.01q_c'$) was 40°.

A comparison of the results of the present tests with previous tests of the same tube at subsonic and transonic speeds showed that the critical angle remained constant at about 63° over a Mach number range of 0.26 to 0.50, decreased to 56° at a Mach number of 1.10, and then decreased rapidly to 40° at a Mach number of 1.61.

Langley Aeronautical Laboratory,
National Advisory Committee for Aeronautics,
Langley Field, Va., December 14, 1953.

REFERENCES

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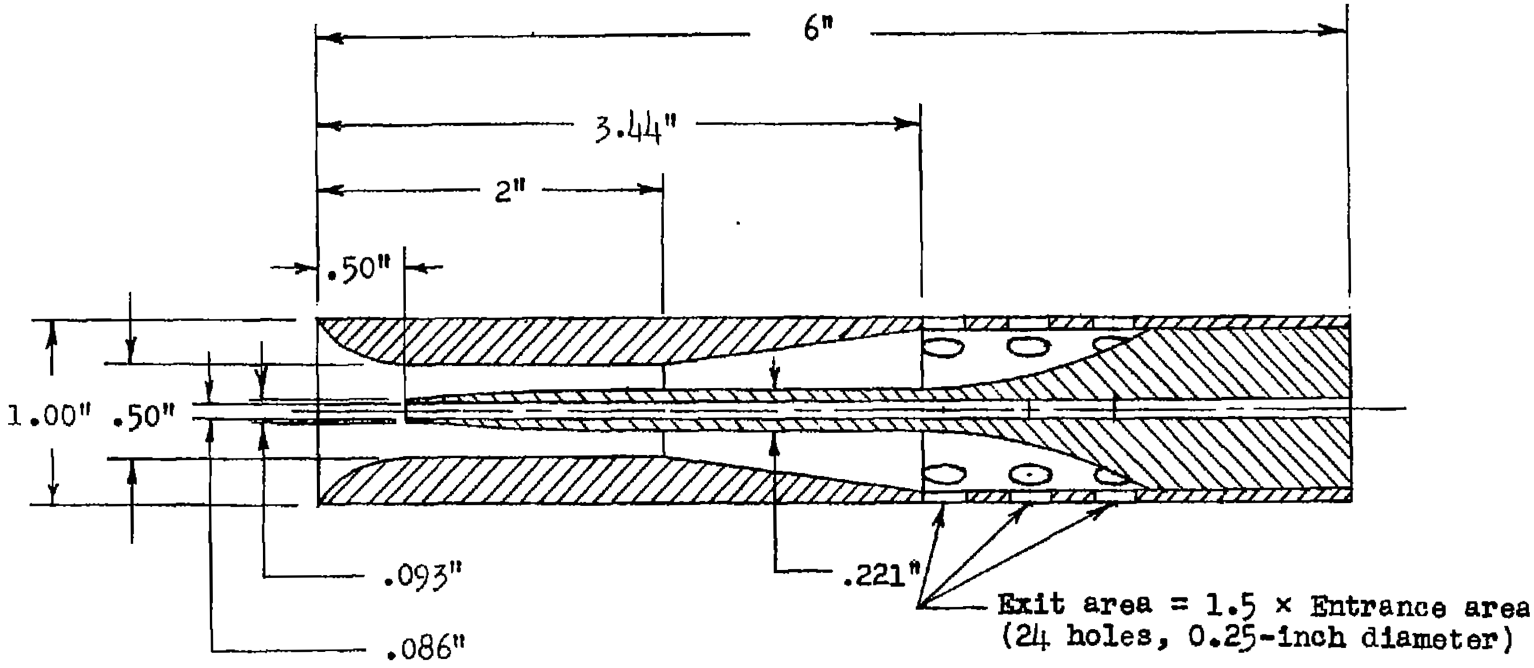

Figure 1.- Section view of shielded total-pressure tube.

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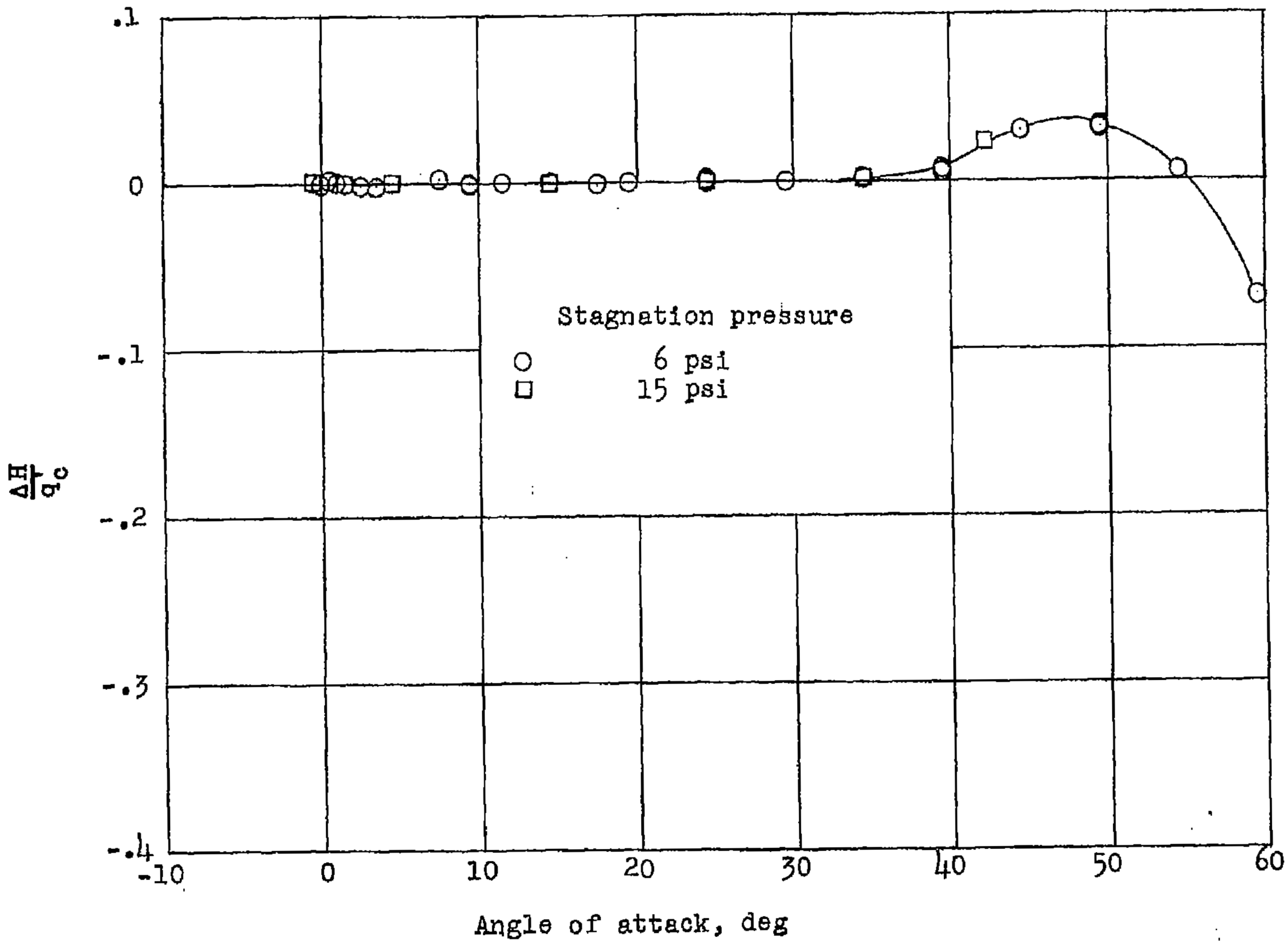

Figure 3.- Variation of total-pressure error with angle of attack.

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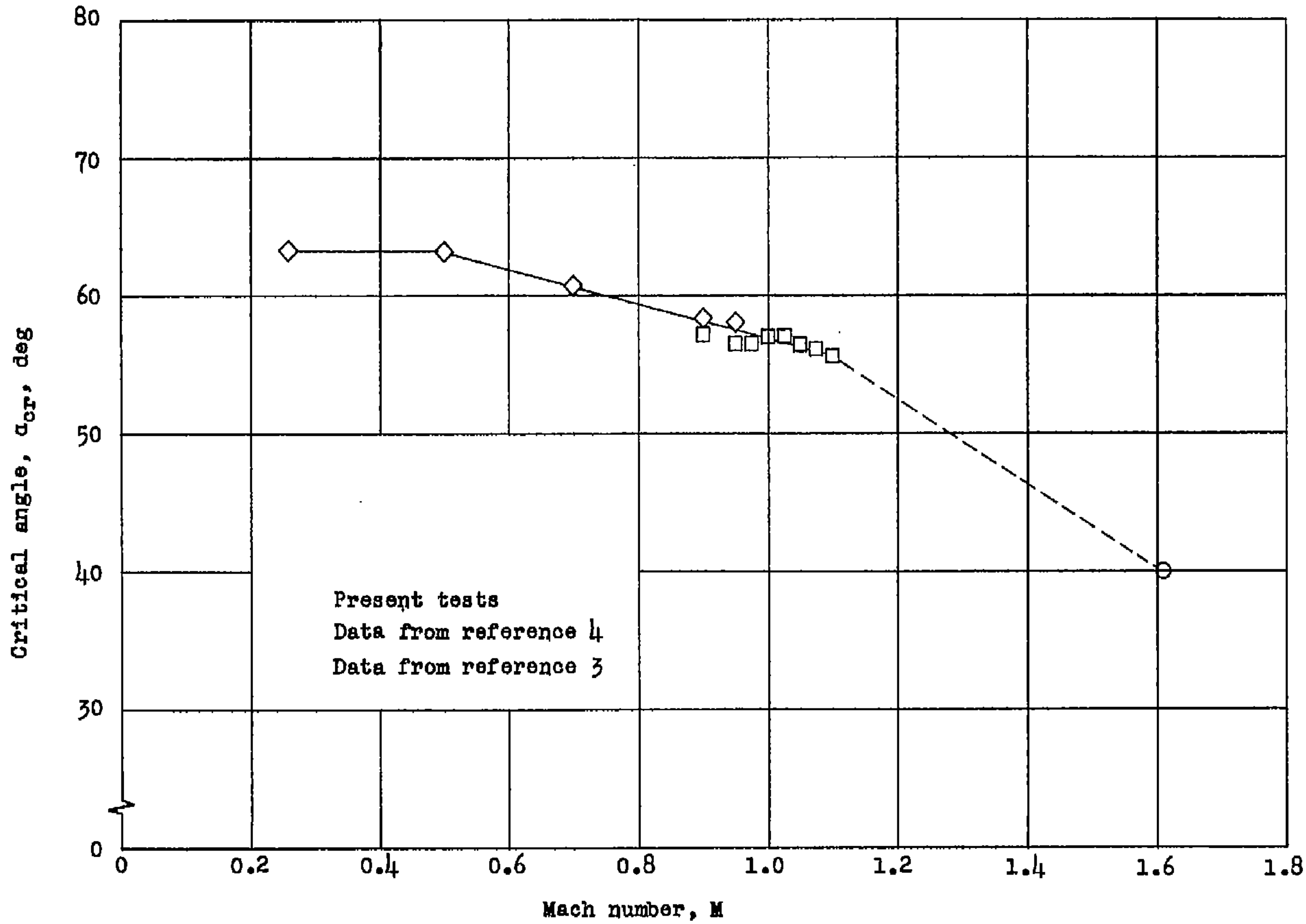

Figure 4.- Variation of critical angle with Mach number.

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